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THE AMES M-50 HELIUM TUNNEL

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16. Abstract

A description of the Ames M-50 helium tunnel and results of calibration tests are presented. Included in the test results are pitot-pressure and total-temperature surveys of the nozzle flow. The pitot-pressure surveys were made using two different throat sections in the nozzle designed to provide test Mach numbers of 40 and 50. The Mach 50 throat was found to have problems with thermal stresses which caused significant variations in test Mach number from run to run, thus rendering it unsuitable for research. However, the Mach 40 throat was found to provide a uniform test core 10.16 cm (4 inches) in diameter and 25.40 cm (10 inches) long with pitot-pressure variations of less than ±5 percent, giving Mach number variations less than ±1.8 percent. Tests on the Mach 40 nozzle indicate a uniform total-temperature distribution through the test core.

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SYMBOLS

M	Mach number
p_o	reservoir or stagnation pressure
p_{t_2}	pitot pressure
r	nozzle radius
r^*	sonic throat radius
T_n	temperature measured on inner wall of nozzle throat
T_o	reservoir or stagnation temperature
T_{t_2}	stagnation temperature downstream of a normal shock
x	longitudinal distance measured downstream from the nozzle throat
y	vertical distance from the nozzle centerline, positive up
z	horizontal distance from the nozzle centerline, positive to the right when looking upstream
δ	boundary-layer thickness
δ^*	boundary-layer displacement thickness

Subscript

t	computed or theoretical value

THE AMES M-50 HELIUM TUNNEL

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SUMMARY

A conventional type wind tunnel capable of producing ultra-high Mach number (40 or greater) flows at unit Reynolds numbers from 20,000 to 32,000 per cm (50,000 to 80,000 per inch) has been developed and built at the Ames Research Center. A description of this facility, known as the Ames M-50 helium tunnel, and the results of calibration tests are presented. Included in the test results are pitot-pressure and total-temperature surveys of the nozzle flow; the pitot-pressure surveys were made using two different throat sections in the nozzle designed to provide test Mach numbers of 40 and 50. The Mach number 50 throat experienced thermal stresses sufficiently high to cause plastic deformation. These deformations caused variations in Mach number from run to run which rendered the nozzle unsuitable as a research tool. However, the Mach number 40 throat was found to provide a stable test core, 10.16 cm (4 inches) in diameter and 25.40 cm (10 inches) long in which pitot-pressure variations were less than ±5 percent giving Mach number variations less than ±1.8 percent. Furthermore, tests on the Mach number 40 nozzle indicate a uniform total-temperature distribution through the test core.

INTRODUCTION

In order to experimentally investigate problems associated with superorbital entry into planetary atmospheres it is desirable to have facilities capable of producing ultra-high Mach number (40 and greater) flows at Reynolds numbers on the order of 10^5 to 10^6. These Mach number and Reynolds number conditions can be obtained in a conventional type wind tunnel using heated helium as a test gas. Such a facility can be designed to have long run times and to operate with reasonably low stagnation temperatures (less than 1111° K (2000° R)) thus lending itself to the use of conventional testing techniques. Furthermore, the properties of ideal-gas helium flow in such a tunnel are well defined and, consequently, test results would be amenable to theoretical correlations. Thus, this facility could be a particularly useful tool for studying viscous effects characteristic of high Mach number flows.

A facility of this type has been developed at the Ames Research Center. The facility was designed to operate at Mach numbers of 40 and 50, and Reynolds numbers compatible with entry conditions for lunar and planetary missions. A description of this facility and its associated equipment is given in this report along with the results of flow calibration tests.

DESCRIPTION OF FACILITY

The Ames M-50 helium tunnel shown in figures 1, 2, and 3 is a blowdown type with an open test section. It is designed to operate with helium as the test gas at temperatures to 1089° K (1960° R) and reservoir pressures to 680 atm (10,000 psia). The nozzle is a contoured, axisymmetric type with two interchangeable throat sections, one designed to provide a nominal Mach number of 40 (Mach 40 throat) and one designed to provide a nominal Mach number of 50 (Mach 50 throat). A steam ejector system maintains the back pressure at less than 0.0034 atm (0.05 psia). The helium is recovered from the steam ejector system, purified and returned to storage for further use. The run time is limited by the power supply for the heater and varies from 3 to 20 minutes, depending on the power required.

Heater

An electrical resistance-type heater shown in figure 4 is used to raise the gas temperature sufficiently high to prevent condensation of the helium flow in the test section. The heating elements are formed of thirty-six 0.794 cm (5/16 in.) o.d. by 0.0764 cm (0.030 in.) wall Inconel-600 tubes. The flow is constrained so that it enters at one end of the heater, flows toward the other end next to the outer structural wall then back between an inner shell and an outer shell, and finally through the resistance tubing where the heating of the gas occurs. This flow pattern provides continuous cooling of the outer pressure shell of the heater.

The heater is capable of operating at tube temperatures of about 1333° K (2400° R); it is equipped with a remote manual control for regulating the power output up to 2 MW which is sufficient to heat the test gas, at the maximum mass flow of 0.91 kg (2 lb) per second, to 1089° K (1960° R).

Pressure Controller

The stagnation pressure in the tunnel may be controlled either manually or by a closed-loop servo system. The servo system establishes the stagnation pressure[1] to within 1 percent of any preset value between 68 and 680 atm (1,000 and 10,000 psia) by a ramp-type increase in pressure. The time required to establish the pressure is 20 to 60 sec depending on the desired operating pressure. Once the pressure is established the fluctuations are controlled to within ±1/2 percent of the operating pressure.

[1]Under present operating conditions the pressure is measured in the back of the heater near the point where the gas enters. However, results from the initial check out of the facility indicate that the difference between the pressure at this point and the pressure at the nozzle entrance is always less than 0.068 atm (1 psia) for the steady-state test conditions.

Nozzle

For the supersonic portion of the nozzle a theoretical inviscid core was obtained by a method-of-characteristics solution developed by Harvard Lomax and Harry E. Bailey at Ames Research Center. A boundary-layer correction was then obtained assuming an adiabatic wall and using a computer program based on the method of Persch and Lee (ref. 1). This program was developed by Robert L. McKenzie at Ames Research Center. Initial core and boundary-layer properties were obtained from numerical calculations for a design stagnation pressure of 680 atm (10,000 psia) and a 30.5 cm (12 in.) diameter test core. The results of these calculations are presented in figure 5 showing the uniform test core, the boundary-layer thickness, δ, and the boundary-layer displacement thickness, δ^*. From these calculations it was evident that for a fully contoured nozzle, the boundary layer would be extremely thick relative to the core size. Furthermore, it was recognized that the computed boundary-layer corrections could easily be in error since these computations represented a very large extrapolation of available information. Consequently, in an effort to reduce the effect of possible errors, the nozzle was terminated short of the fully contoured length, at the point where the uniform core diameter was 10.16 cm (4 in.). At this point the nozzle radius was about 35.56 cm (14 in.) with the boundary-layer thickness approximately 1.5 times the inviscid core radius.

The subsonic portion of the nozzle (see insert of fig. 5) was a 27° half-angle conic section faired to the supersonic portion by a quadratic equation that matched the radius, the slope, and the curvature at the junction point.

A full nozzle with the theoretical contour for a Mach number 50 core was built so that the throat section, up to 30.5 cm (12 in.) into the supersonic region, could be removed. A second throat section was built using the theoretical contour for a Mach number 40 core. Moving the throat 2.032 cm (0.8 in.) downstream, and increasing the throat radius (see insert of fig. 5) made it possible to match this throat section to the Mach 50 nozzle. The resulting nozzle shape with the Mach 40 throat inserted was very close to the theoretical contour for Mach number 40 for the entire length of the nozzle. Except for the change in the core Mach number, the calculated flow distribution for the nozzle with the Mach 40 throat is nearly the same as that presented for the Mach 50 throat shown in figure 5.

A sketch of a throat section is shown in figure 6. The throat sections are built from 410 stainless steel and are equipped with a heating coil around the outside so that they may be heated before the run. The reasons for preheating the nozzle throat are:

1. To reduce the heat transfer to the walls in the subsonic and lower supersonic portion of the nozzle, thus reducing possible temperature gradients in the test core.

2. To reduce the thermal expansion at the nozzle throat during a run, thus reducing the variations in test core characteristics with time.

3. To provide a more uniform temperature distribution in the nozzle wall. This will reduce the thermal stress at the throat and consequently reduce the possibility that plastic deformation will occur.

It was calculated that a temperature difference of 222° K (400° R) between the inner and outer walls would cause a thermal stress of over 551 MN/m^2 (80,000 psi). Since inner wall temperatures typically reach 666° to 888° K (1200° to 1600° R) in this region, differences of 222° K (400° R) or greater could easily occur if no prior heating were applied. Hence it was imperative that the thermal stresses be reduced since stresses of this magnitude would cause plastic deformation which would change tunnel flow characteristics.

Boundary-Layer Injector System

An injector system similar to that described in reference 2 is used to increase momentum in the tunnel boundary layer for the purpose of preventing disturbances created by the model and diffuser from feeding upstream and perturbing the inviscid flow. This injector system, shown in figure 7, is a secondary annular nozzle around the main nozzle exit. It is formed by a contoured diverging section with 48 small converging sections feeding it at equally spaced intervals. The flow from the injector system is helium at about Mach number 20 in a slightly underexpanded condition. The system operates at reservoir pressures from 13.6 to 68 atm (200 to 1000 psia) and typically has a mass flow comparable to that of the main nozzle.

Test Chamber and Diffuser

The test chamber shown schematically in figure 3 is a cubical box with sides approximately 3.048 m (10 ft) long. Circular windows, 30.48 cm (12 in.) in diameter, are located on each of the four sides parallel to the stream axis. The nozzle and the diffuser extend through the wall of the test chamber, with nozzle fixed to the wall and the diffuser mounted with a sliding fit so that the axial location of its entrance can be varied from 15.24 to 66.1 cm (6 to 26 in.) from nozzle exit (see fig. 3).

The facility was initially equipped with the converging-diverging type diffuser indicated by solid lines in figure 3. Later the diffuser was modified by extending the constant diameter throat for an additional 8 feet (indicated by dashed lines in fig. 3) and dumping the flow directly into the 1.219 m (4 ft) diameter connecting pipe without benefit of a divergent section.

Model Support System

The model support (fig. 8) is a hydraulically operated, quick-insert mechanism designed for use with sting-mounted models. The system provides freedon of motion in three directions, vertical (perpendicular to the tunnel axis), longitudinal (parallel to the tunnel axis), and rotational (or angle-of-attack motion). The vertical motion is generated by a servo-controlled hydraulic cylinder capable of inserting a model into the test stream in about 0.7 second. Angles of attack from -15° to +45° are obtained by rotating the circular mounting ring to which the strut is connected. The strut provides a fixed center of rotation at the center of the ring. This motion is also generated by a servo-controlled hydraulic cylinder. The longitudinal motion of the system for distances up to 43.2 cm (17 in.) is achieved by use of an air-hydraulic cylinder.

All motions of the model support are remotely controlled by a system that provides for manual control of the longitudinal position, and either manual or automatic control of the vertical position and angle of attack. Under automatic control, the system will position the model and provide a pulse to operate the data recording system for 25 discrete angles of attack. Stepping between the various angles of attack may be accomplished at either fixed time intervals or by a return pulse from the data recording system.

Data Recording System

A Beckman 210 High-Speed Data Recording System is used for data acquisition. This system accepts analog dc voltage inputs, converts these inputs to binary coded decimal digits, and then records the information on a 1.27 cm (1/2 in.) magnetic tape. The system will record 100 channels of data. Of these channels 90 will accept analog inputs as low as 1 mV full scale and 10 require a 5 V input for full-scale readout. An unusual feature of the system is a logical control circuit that analyzes one channel of data and can be used to delay data recording until the time stability of that channel is within preset limits.

INSTRUMENTATION FOR CALIBRATION TESTS

This section contains a description of the survey rakes and support equipment used to calibrate the test stream.

Pitot-Pressure Rake

The pitot survey rake shown in figure 9(a) had 15 probes, each with an outside diameter of 0.3175 cm (0.125 in.) and an inside diameter of 0.1067 cm (0.042 in.). The probes were located 1.27 cm (0.5 in.) apart for 7.62 cm (3 in.) on either side of the rake center and the two outer probes were located 10.16 cm (4 in.) from the centerline. (See fig. 9(a).) Pressures were measured by strain-gage transducers calibrated for the ranges encountered in these tests (maximum pressures from 0.034 to 0.068 atm (0.5 to 1.0 psia)). The transducers were found to repeat readings to within ±1 percent of the measured value. The tunnel Mach number was determined from the pitot-pressure measurements, assuming isentropic flow and using the real-gas correction to the pressure ratio, p_{t_2}/p_o, given by reference 3. This correction accounts for the compressibility effects of helium at the high reservoir pressures used in this facility.

Total-Temperature Rake

The total-temperature rake consisted of 5 probes located 2.54 cm (1 in.) apart as shown in figure 9(b). Triple-shielded probes, 0.635 cm (0.25 in.) in diameter, were used to minimize radiation losses. The probe sensing element was a platinum-platinum 10-percent rhodium thermocouple junction.

RESULTS AND DISCUSSION

Effect of Diffuser Location on the Flow

The effect of the diffuser entrance location on the flow was determined by surveys made 1.27 cm (0.5 in.) downstream of the nozzle exit with the diffuser at various longitudinal positions from 15.24 to 66.1 cm (6 to 26 in.) from the exit. Comparisons of these surveys revealed that the location of the diffuser had no effect on the pitot-pressure distribution at the lower reservoir pressures (136 atm (2000 psi) or less). However, as the reservoir pressure was increased for each diffuser location, a pressure was reached at which the tunnel choked, resulting in a breakdown in isentropic flow. The pressure at which choking occurred was found to depend on diffuser location. For an open-jet length of 15.24 cm (6 in.), choking occurred at a relatively low pressure (about 136 atm (2000 psia) for the Mach 40 throat and about 272 atm (4000 psia) for the Mach 50 throat); however, as the diffuser entrance was moved downstream, the pressure at which choking occurred increased until it reached a maximum for an open jet length of about 61 cm (24 in.). With the diffuser at this location the maximum reservoir pressure at which the tunnel could be operated was about 306 atm (4500 psia) for the Mach 40 throat, and about 612 atm (9000 psia) for the Mach 50 throat. After these pressure limitations were observed, the diffuser throat section was extended as described earlier (see section on Test Chamber and Diffuser). It was hoped that this modification would increase the efficiency of the diffuser and thus provide a capability of operating at higher reservoir pressures. However, tests indicate that this modification did not change the tunnel operating characteristics.

Mach 40 Nozzle Throat

Pitot-pressure surveys- Pitot-pressure profiles for the Mach 40 throat section at a station 1.27 cm (0.5 in.) from the nozzle exit are presented in figure 10. Also shown in the right-hand side of this figure are the corresponding Mach numbers for the central region of the tunnel where the assumption of isentropic flow applies. For reservoir pressures above 136 atm (2000 psia) the pressure distribution in the tunnel is very erratic. This is probably the result of an overexpansion of the flow and the subsequent creation at oblique shock waves.

For pressures from 102 to 136 atm (1500 to 2000 psia) the core size and Mach number appear to be near the design core diameter of 10.16 cm (4 in.)

and Mach number of 40. These pressures, however, are considerably below the design operating pressure of 680 atm (10,000 psia). This difference may be explained by the data in figure 11. In this figure experimental data from reference 4, which reported results of a detailed study of the nozzle boundary layer in this tunnel, are compared with the theoretical values computed for the nozzle design. The experimental data were obtained at p_o = 108.8 atm (1600 psia) and the theoretical values computed for a p_o = 680 atm (10,000 psia). The agreement between the theoretical and experimental displacement thickness is very good as would be expected because of the agreement between experimental and theoretical Mach numbers. However, the measured boundary-layer thickness is much greater than the theoretical thickness as might be expected because of the lower pressure. Thus it appears that the major deficiency in the theoretical computations for the boundary layer is that they provided too large a ratio of δ^*/δ. Consequently, the design program overcorrected for boundary-layer thickness, causing a much greater effective area ratio than the design value and a relatively poor nozzle contour at the design operating pressure. However, it appears that lowering the operating pressure increased the boundary-layer thickness until at a reservoir pressure of about 102 to 136 atm (1500 to 2000 psia), the increased thickness compensated for the error in the design computations. The resulting nozzle is reasonably well contoured for operation at Mach number 40 and these pressures.

Additional profiles of pitot pressure are shown in figures 12 and 13 for reservoir pressures between 102 and 136 atm (1500 and 2000 psia) at 12.7 and 24.1 cm (5 and 9.5 in.) from the nozzle exit. From the data in figures 10, 12, and 13, it is apparent that for reservoir pressures from 102 to 136 atm (1500 to 2000 psia), the region with a reasonably uniform pressure distribution (i.e., the uniform core) is about 10.16 cm (4 in.) in diameter for the three stations surveyed. Within this region the pitot-pressure variations are less than ±5 percent giving Mach number variations of less than ±1.8 percent. The parameters that define the core Mach number are wall temperature of the nozzle throat and reservoir pressure. In figure 14 the mean core Mach number is plotted versus stagnation pressure for three longitudinal stations at nozzle temperatures of 622° and 656° K (1120° and 1180° R). From these plots it is evident that no measurable longitudinal Mach number gradient exists in the tunnel since the scatter of the data at any given station is as great as the differences between the data obtained at the various longitudinal locations. At pressures above about 136 atm (2000 psia), the variation in Mach number from run to run and the lateral gradients in the tunnel became much larger than those indicated for the lower pressures.

In figure 15 variations in the mean core Mach number with nozzle temperature are shown for various reservoir pressures. The lines shown in the figure are straight-line fairings of the data. In general, the fairings represent the measurements to an accuracy within the data scatter of ±1.8 percent of the Mach number. The heat transfer causes the nozzle temperature to reach near adiabatic wall conditions (75 to 80 percent at the stagnation temperature) in about 2 minutes and to be reasonably stable after that. Since run times may be as long as 20 minutes, most of the data are obtained with the nozzle temperatures near adiabatic wall conditions. For these temperatures the test-stream properties for the tunnel with the Mach 40 throat are summarized in table 1.

Since the boundary layer of a tunnel such as this is primarily hypersonic, a disturbance at the tunnel wall within the last few feet of the nozzle exit should have no effect on the core flow of the tunnel test section since the entire test core would be ahead of the Mach line originating at the wall within this region. To verify this, approximately the last third (132 of 361 cm (52 of 142 in.)) of the divergent section of the nozzle was removed and a pitot survey of the flow was made. The comparison of the pressure profile of the core area measured at station 1.27 cm (0.5 in.) with and without the end section of the nozzle is given in figure 16. These data were obtained without the boundary-layer injectors operating. As was expected, removing the nozzle end had no effect on the flow characteristics near the center of the test core but did cause an apparent thickening of the boundary layer of about 2.54 cm (1 in.).

Total temperature surveys- A typical temperature distribution for the Mach 40 nozzle is shown in figure 17. The values presented are actual measurements with no corrections for possible density or Mach number effects. The data indicate that the temperature distribution within the test core is nearly constant at about 2 percent below the reservoir temperature. The variations between the core temperature and the temperature indicated in the lower Mach number flow surrounding the test core are about 3 percent.

Effect of boundary-layer injectors- No systematic tests have been conducted to determine the effectiveness of the boundary-layer injectors but some information has been obtained from various research projects conducted in the tunnel. With smaller models (3.81 cm (1-1/2 in.) diameter or less) the operation of the injectors makes no difference on the tunnel flow except at high pressure test conditions. At these test conditions the mass flow from the injectors increases the load on the steam ejectors used to pump the flow from the tunnel and causes the downstream pressure to rise. Consequently, the tunnel chokes at a lower reservoir pressure. This is not a serious problem for this tunnel since the pressure at which choking occurs is still above the pressure where flow disturbances are encountered in the core (above 136 atm (2000 psia)).

The boundary-layer injectors have demonstrated a capability of decreasing the effects of model size on tunnel flow. For example, in tests of a 5.08 cm (2 in.) diameter Apollo type model and a 15° wedge model, which was 17.77 cm (7 in.) wide and had a 0.670 cm (0.264 in.) diameter cylindrical bluntness, the tunnel would choke without the boundary-layer injectors operating. However, when the boundary-layer injectors were started before the models were inserted into the test region, the tunnel did not choke. As a result of these observations it is standard procedure to operate the boundary-layer injectors at all times to limit possible model interference effects.

Mach 50 Nozzle Throat

Pitot-pressure profiles with the corresponding Mach number variations for the nozzle with the Mach 50 throat are shown in figure 18. During the first run with this throat (run A in fig. 18), the pitot pressures indicated fairly reasonable distribution through the center or core region of the tunnel. However, the indicated Mach number in this region was much greater than the design value, probably because of the error in the theoretical displacement thickness which resulted in an overcorrection for the boundary-layer thickness, as discussed previously in the discussion of Mach 40 nozzle results. Additional surveys of the flow in the nozzle with the Mach 50 throat indicated that the results could not be repeated from run to run. This variation is demonstrated by the profiles for the seventh run with this nozzle throat (run B in figs. 18(b) and 18(c)). For this run the pitot-pressure distribution is seen to be very different from that for the first run. After this run, it was found that the throat diameter had changed from 0.363 to about 0.345 cm (143 to 136 in.).

This deformation apparently resulted from thermal gradients in the nozzle wall, indicating that preheating of the nozzle was not as effective as had been initially expected. Because of conduction and convection losses, the outer wall of the nozzle could be preheated to only about 611° K (1100° R) at which time the inner surface was about 444° K (800° R). During a typical run, the inner surface would reach 778° to 833° K (1400° to 1500° R), thus resulting in temperature differences of 167° to 222° K (300° to 400° R). These temperature differences caused very high thermal stresses which apparently resulted in a small amount of plastic deformation in the nozzle throat area. The change in throat size cannot directly account for the changes in Mach number encountered in the tests. However, the deformation of the throat is concentrated in a very short region of the nozzle, and because of this, the wall slopes are significantly affected. These changes in wall slopes apparently affected the relative contour of the nozzle to the extent that oblique shocks were induced in the nozzle which in turn produced the indicated Mach number change. Because of this deformation problem the Mach 50 throat is considered unsuitable for use without modification.

CONCLUSIONS

Calibration tests in the Ames M-50 helium tunnel have revealed the following information:

1. The Mach 40 throat was found to provide an inviscid test core approximately 10.16 cm (4 in.) in diameter and 25.4 cm (10 in.) long in which the pitot-pressure variation was less than ±5 percent and the Mach number variation was less than ±1.8 percent.

2. For this core the Mach number varied from 39 to 48 and depended on the reservoir pressure and wall temperature of the nozzle throat.

3. A reasonable sized uniform core was obtained for reservoir pressures from 102 to 136 atm (1500 to 2000 psia).

4. The boundary-layer corrections used in the nozzle design provided a ratio of boundary-layer displacement thickness to total boundary-layer thickness which was apparently too large. Consequently, the calculations tend to overcompensate for the boundary-layer thickness.

5. Total-temperature surveys of the flow revealed no significant temperature gradients within the test core.

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REFERENCES

1. Persch, J.; and Lee, R. E.: A Method for Calculating Turbulent Boundary-Layer Development in Supersonic and Hypersonic Nozzles Including the Effects of Heat Transfer. NAVORD Rep. 4200, June 1956.

2. McDevitt, John B.: A Study of Gas Injection in the Boundary Layer of a Hypersonic Wind Tunnel to Extend the Useful Operating Range. NASA TN D-2935, 1965.

3. Erickson, Wayne, D.: An Extension of Estimated Hypersonic Flow Parameters for Helium as a Real Gas. NASA TN D-1632, 1963.

4. Kemp, Joseph H., Jr.; and Sreekanth, A. K.: Preliminary Results From an Experimental Investigation of Nozzle Wall Boundary Layers at Mach Numbers Ranging From 27 to 47. AIAA paper 69-686, June 1969.

TABLE 1.- TEST STREAM PROPERTIES FOR THE M-50 HELIUM TUNNEL WITH THE MACH 40 THROAT

Reservoir Conditions	
Pressure	102 to 136 atm (1500 to 2000 psia)
Temperature	555° to 1056° K (1000° to 1900° R)
Stream Conditions	
Pitot pressure	0.0408 to 0.0476 atm (0.6 to 0.7 psia)
Dynamic pressure	0.0204 to 0.0272 atm (0.3 to 0.4 psia)
Static pressure	0.0000136 to 0.0000204 atm (0.0002 to 0.0003 psia)
Static temperature	1.334° to 2.00° K (2.4° to 3.6° R)
Density	3.67×10^{-4} to 5.67×10^{-4} kg/m^3 (0.7×10^{-6} to 1.1×10^{-6} slug/ft^3)
Velocity	2440 to 3350 m/sec (8,000 to 11,000 ft/sec)
Mach number	40 to 43
Reynolds number	20,000 to 32,000/cm (50,000 to 80,000/in.)
Test Core	
Diameter	10.16 cm (4 in.)
Length	25.4 cm (10 in.)
Mach number variation	±1.8 percent
Dynamic-pressure variation	±5 percent
Test Time	20 min

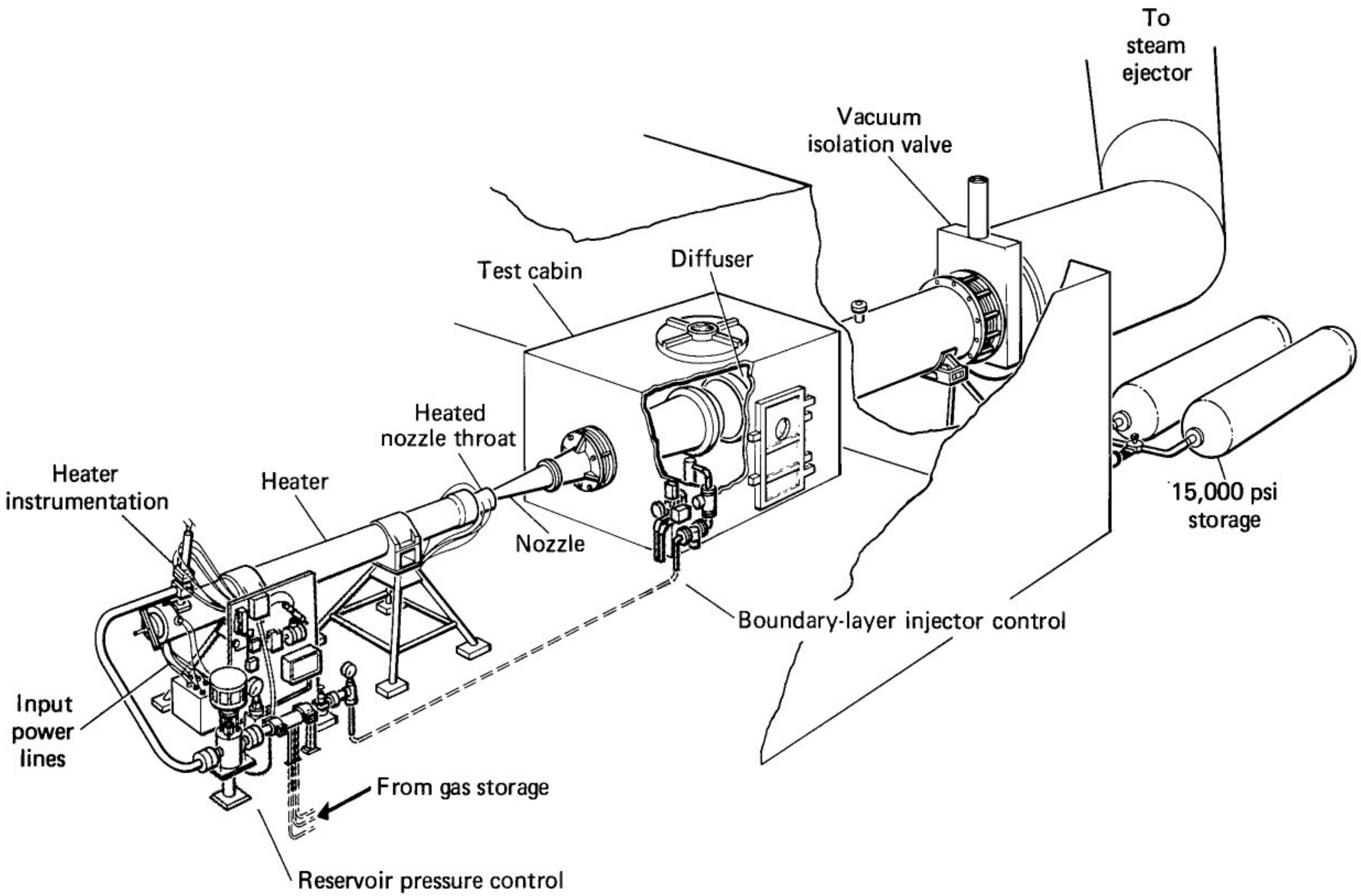


Figure 1.- Schematic diagram of the M-50 helium tunnel.

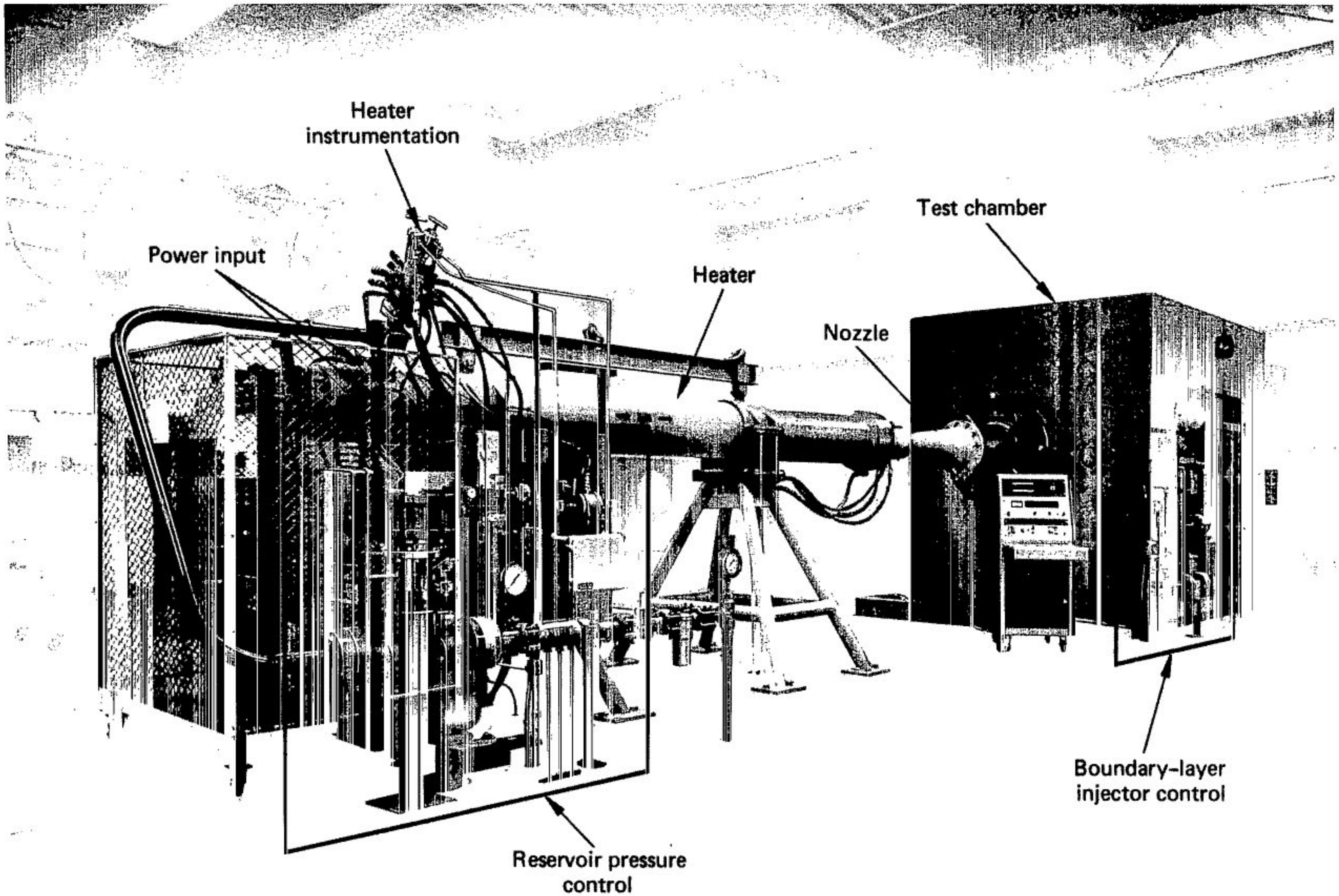


Figure 2.- The M-50 helium tunnel.

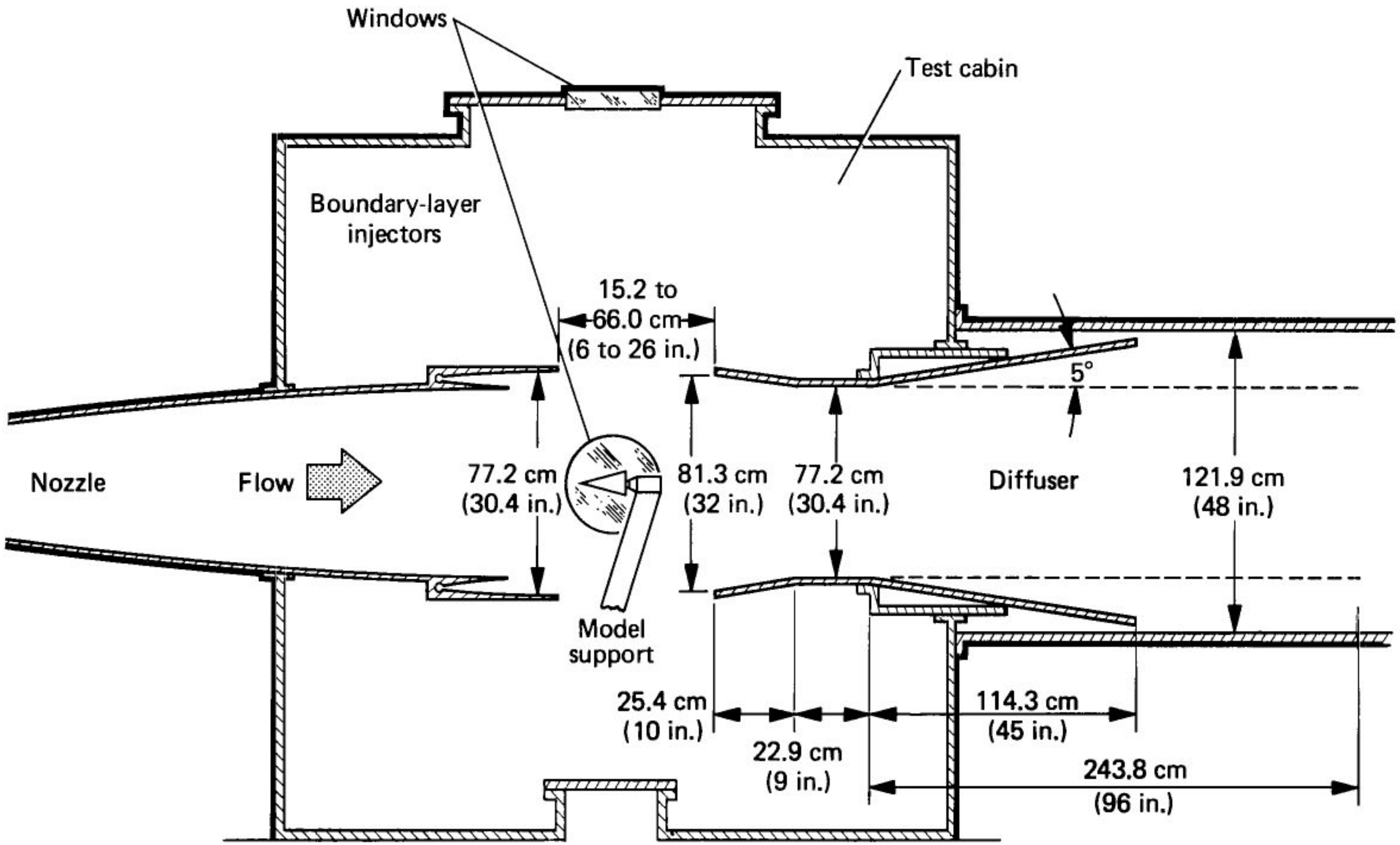


Figure 3.- Schematic drawing of test chamber.

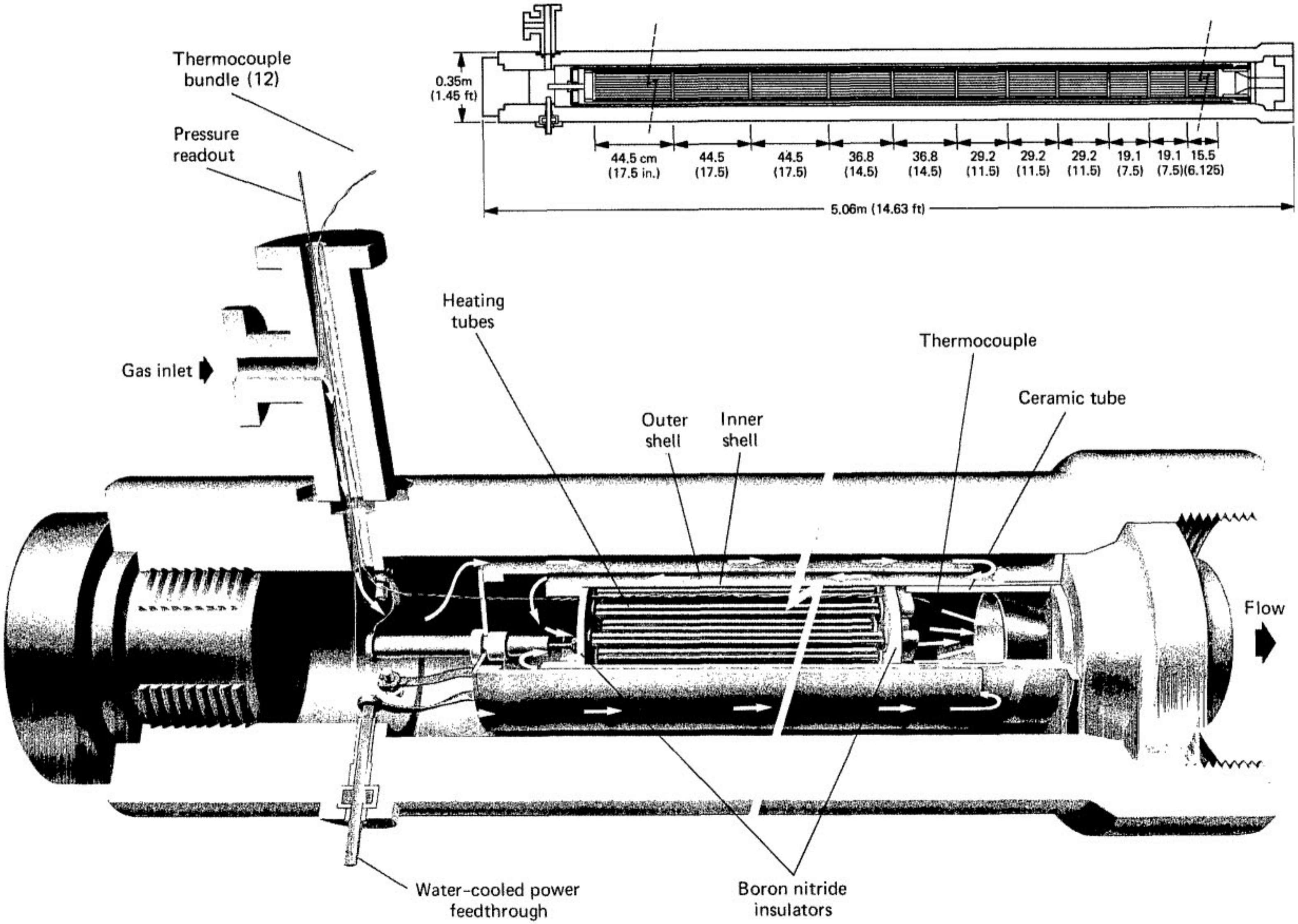


Figure 4.- Schematic drawing of resistance type electrical heater for Ames M-50 helium tunnel.

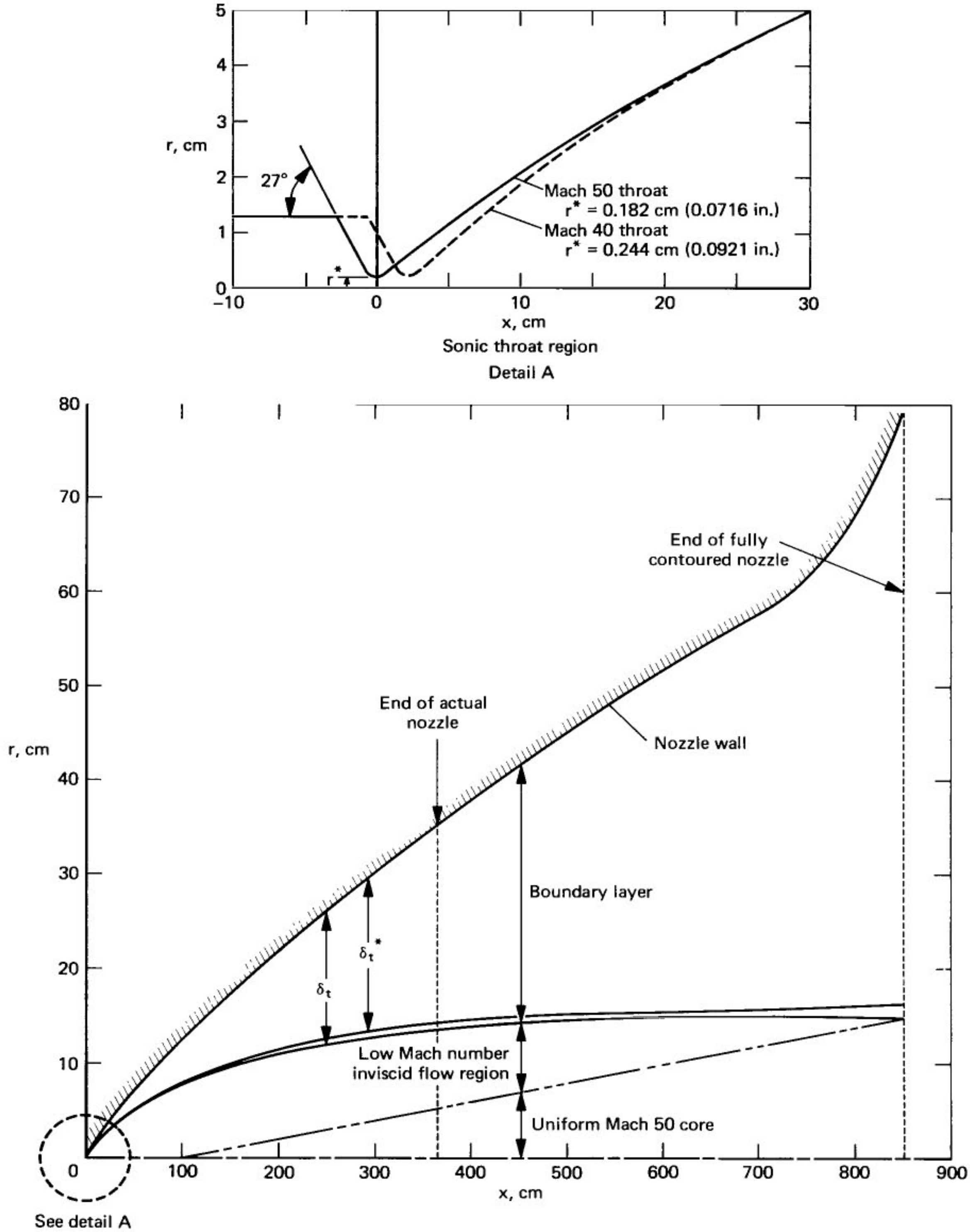


Figure 5.- Estimated flow regimes for Mach 50 nozzle; p_o = 10,000 psia, T_o = 2,000° R.

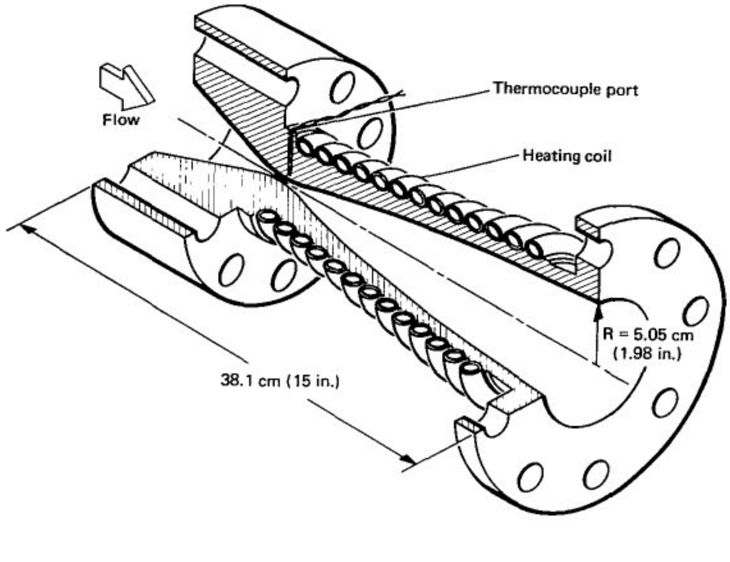


Figure 6.- Diagram of nozzle throat section.

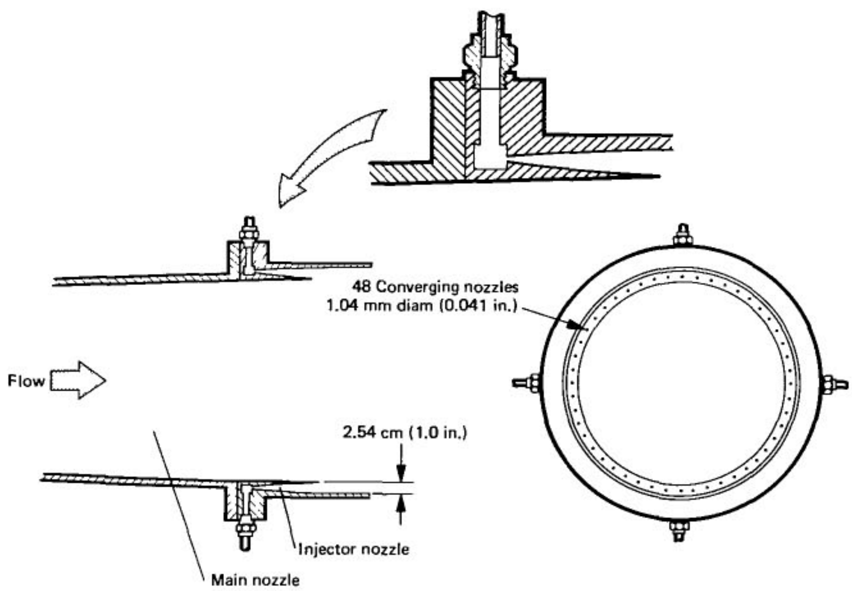


Figure 7.- Schematic drawing of boundary-layer injector system.

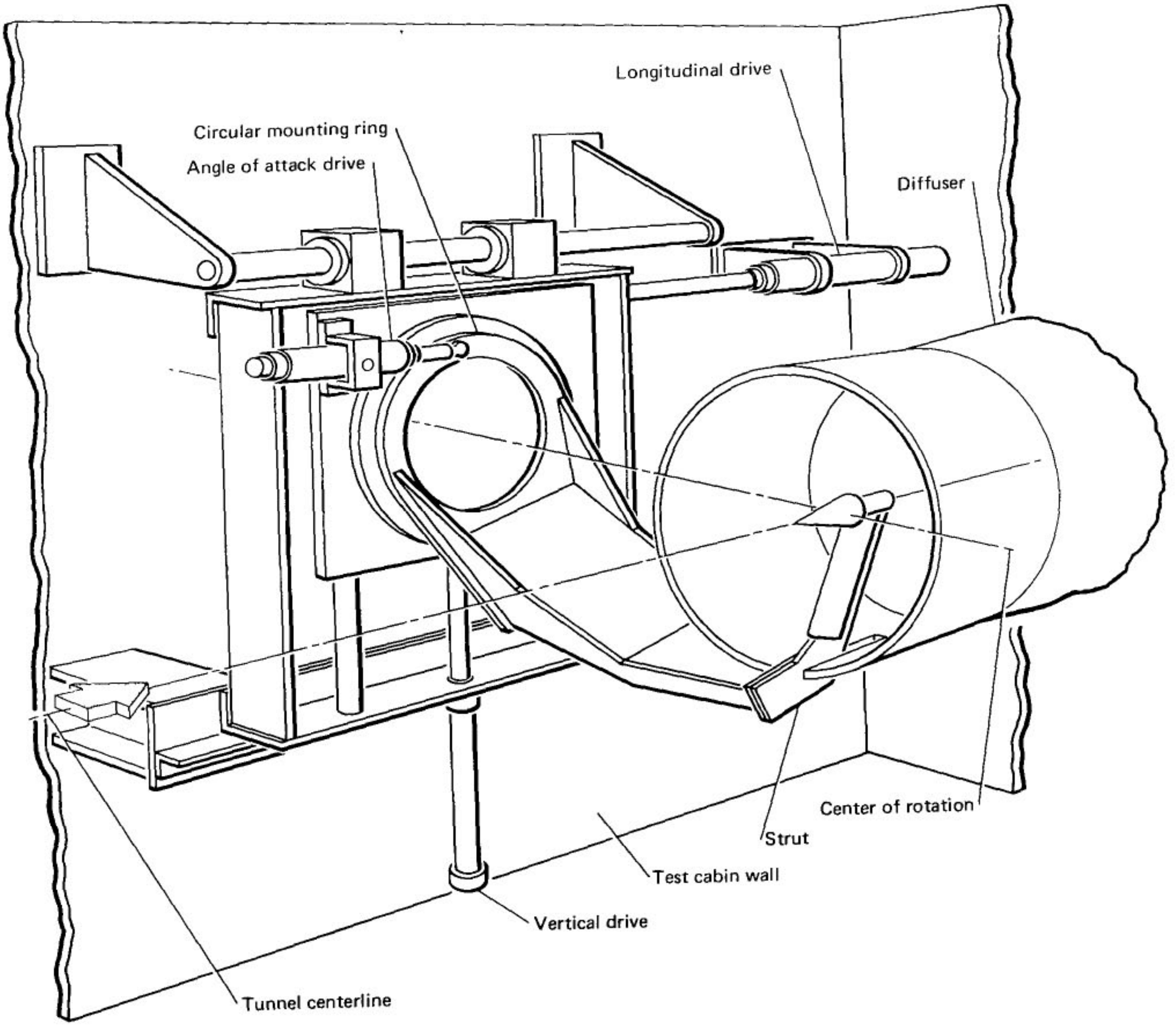
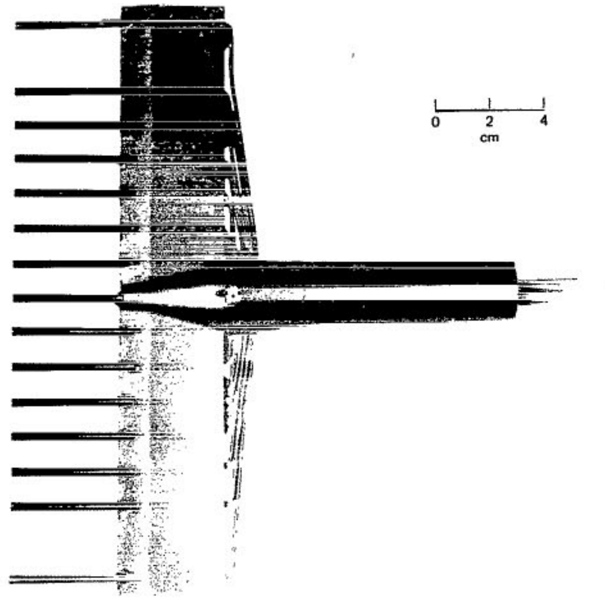
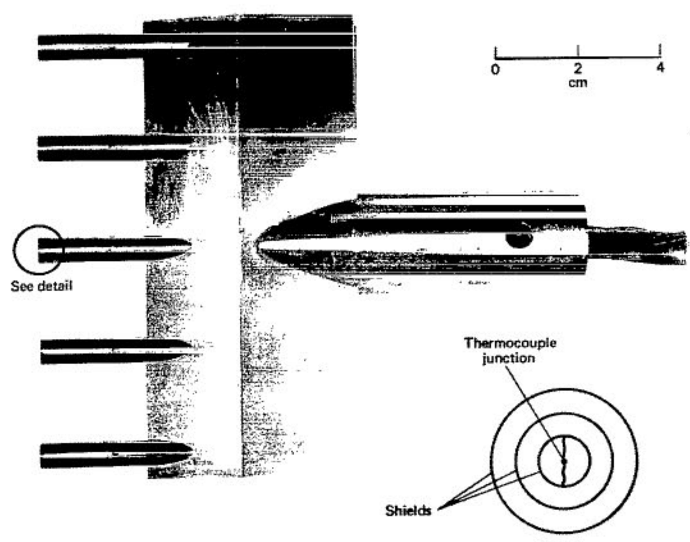


Figure 8.- Schematic drawing of the M-50 model support system.



(a) Pitot pressure rake.



(b) Total temperature rake.

Figure 9.- The calibration rakes.

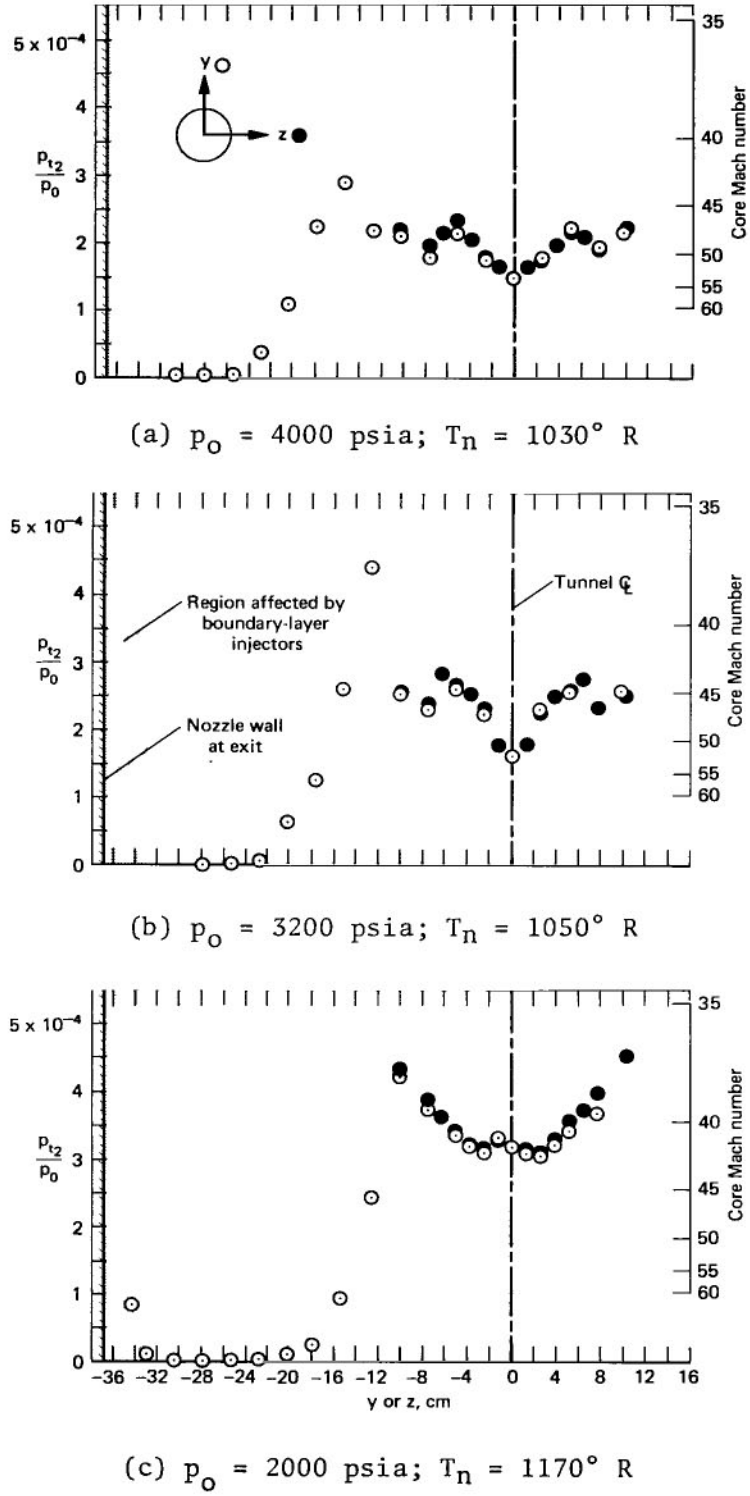
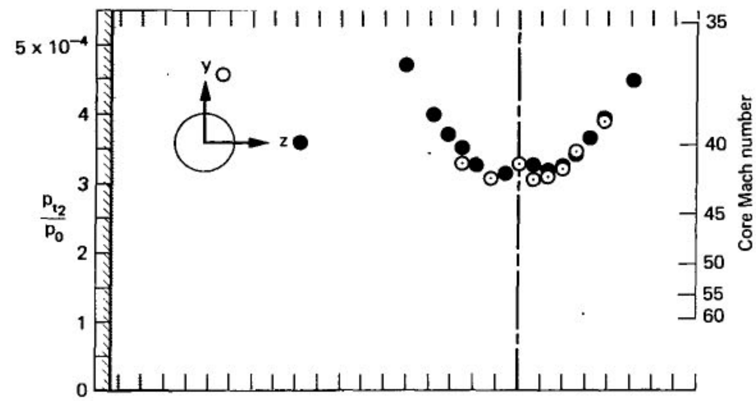
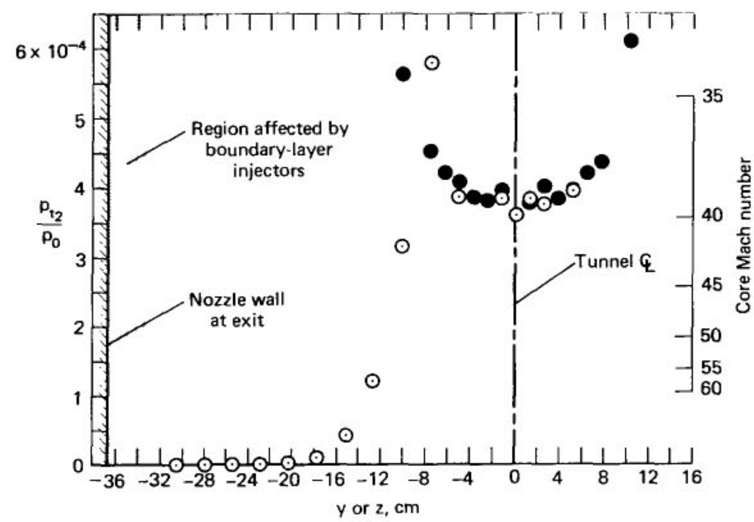


Figure 10.- Measured pressure distribution for the Mach 40 throat at a station 0.5 inch downstream of the nozzle exit.



(d) p_o = 1900 psia; T_n = 1140° R



(e) p_o = 1500 psia; T_n = 1190° R

Figure 10.- Concluded.

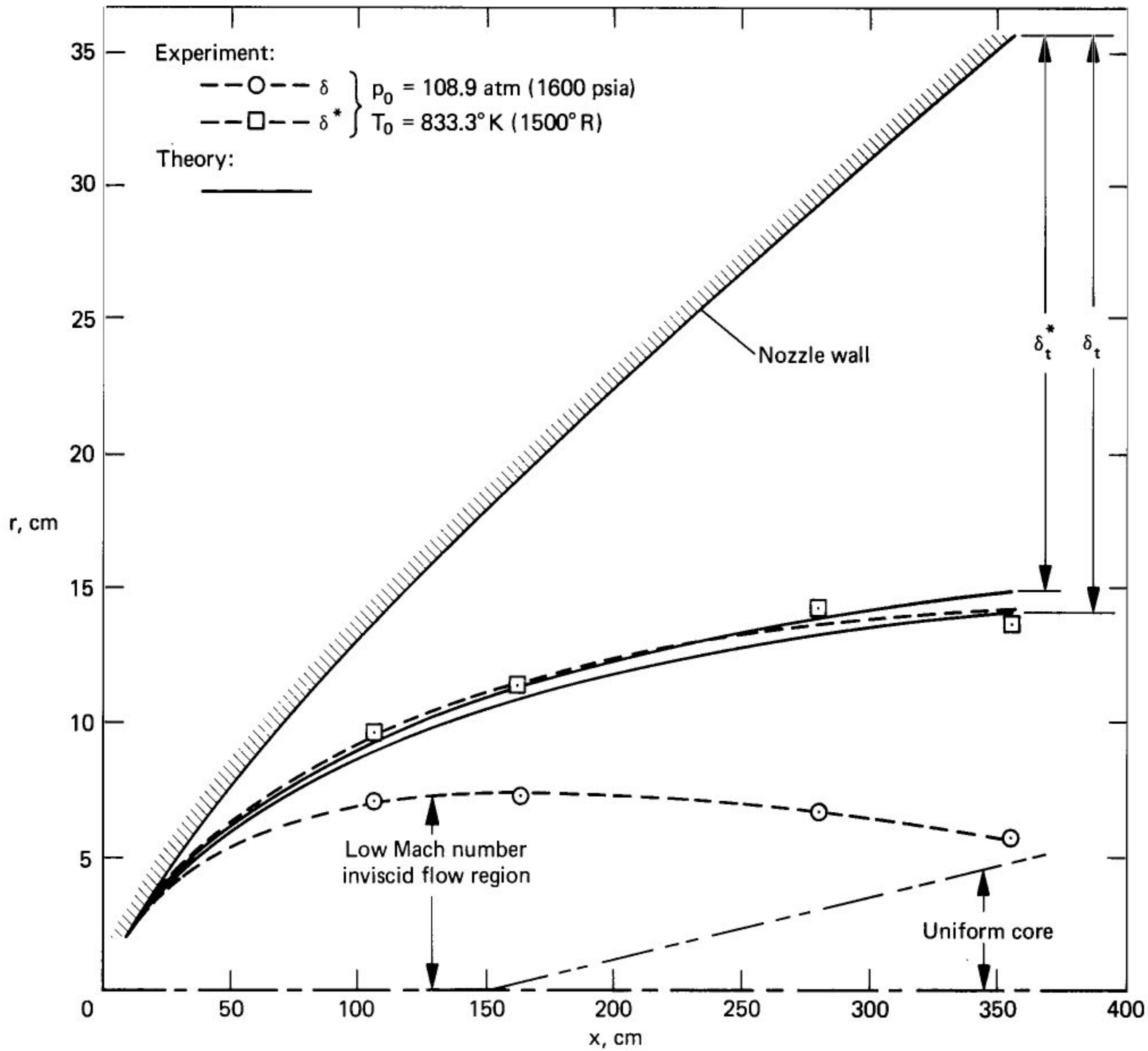
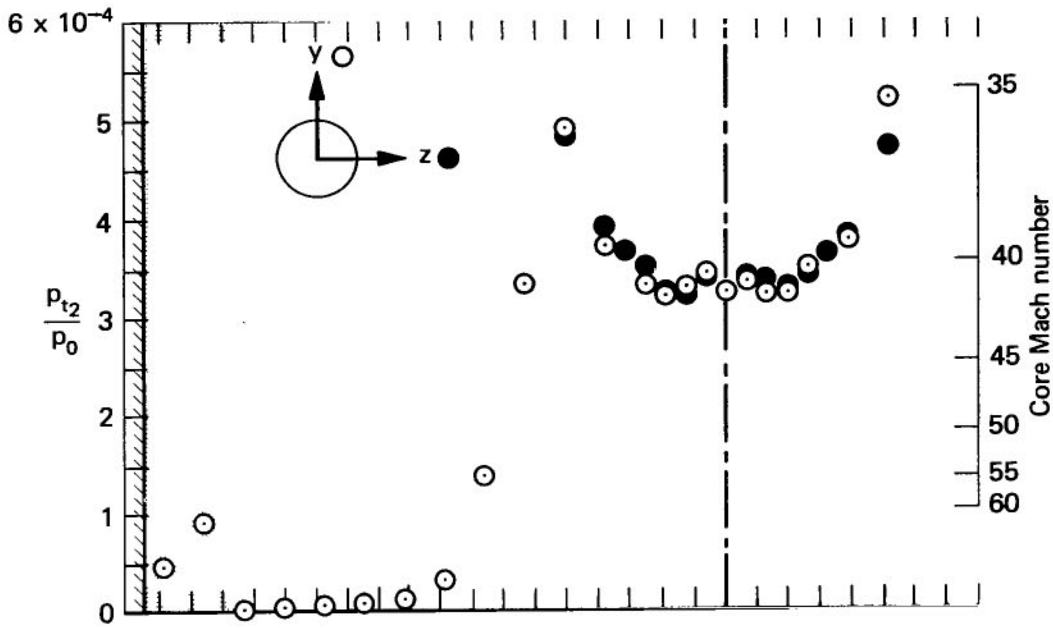
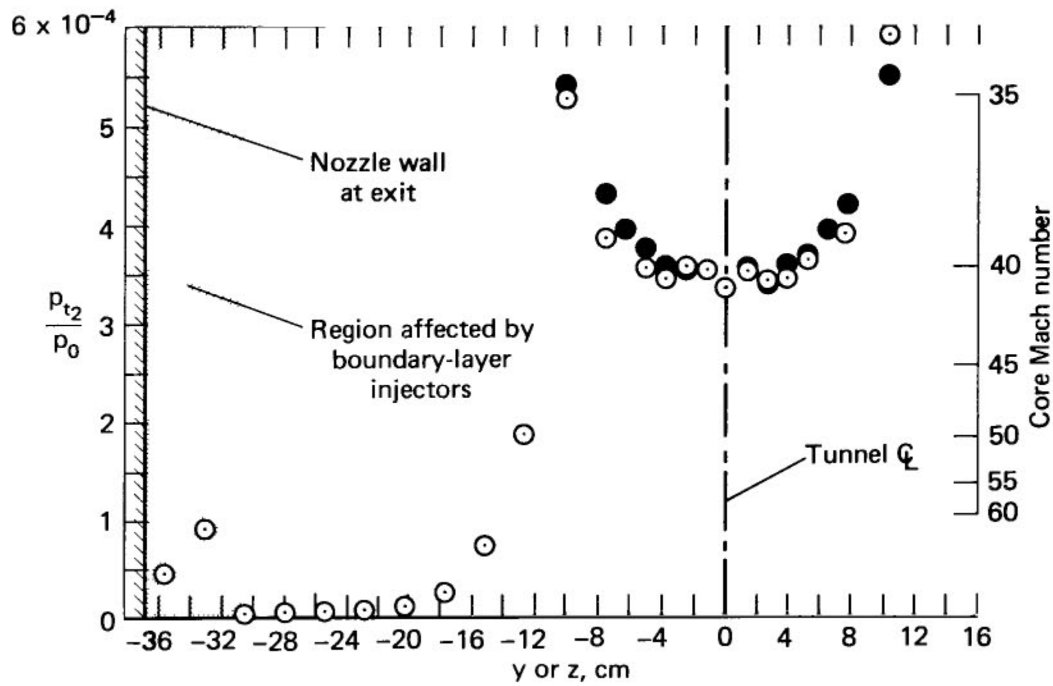


Figure 11.- Boundary-layer thicknesses along the M-50 nozzle.

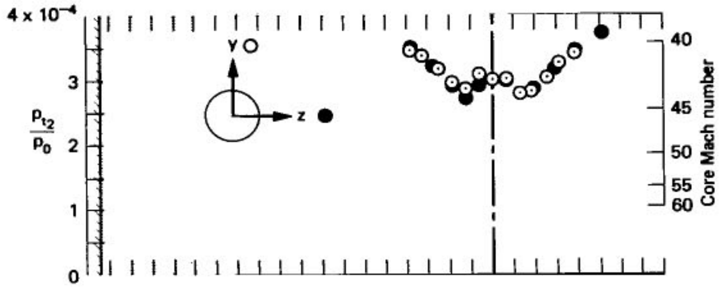


(a) p_o = 1990 psia; T_n = 1210° R

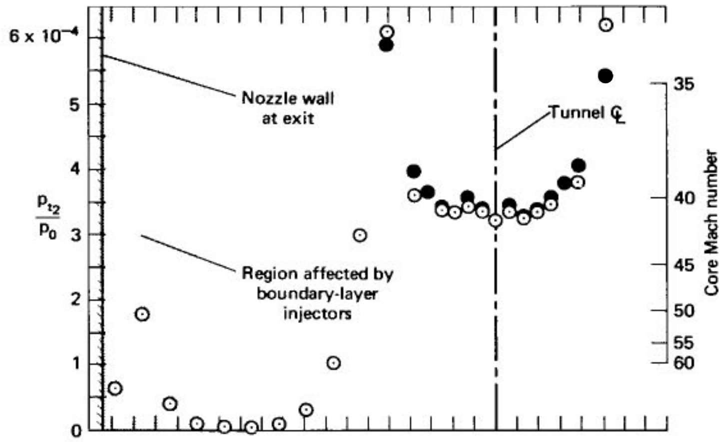


(b) p_o = 1850 psia; T_n = 1220° R

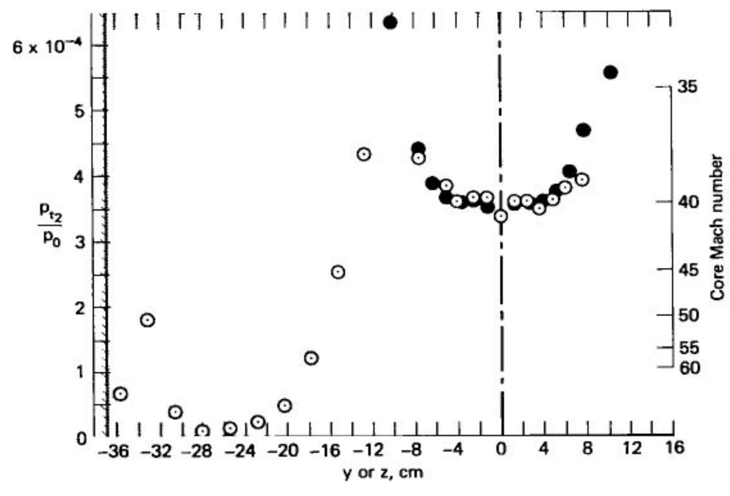
Figure 12.- Measured pressure distribution for the Mach-40 throat at a station 5 inches downstream of the nozzle exit.



(a) p_o = 2070 psia; T_n = 1170° R

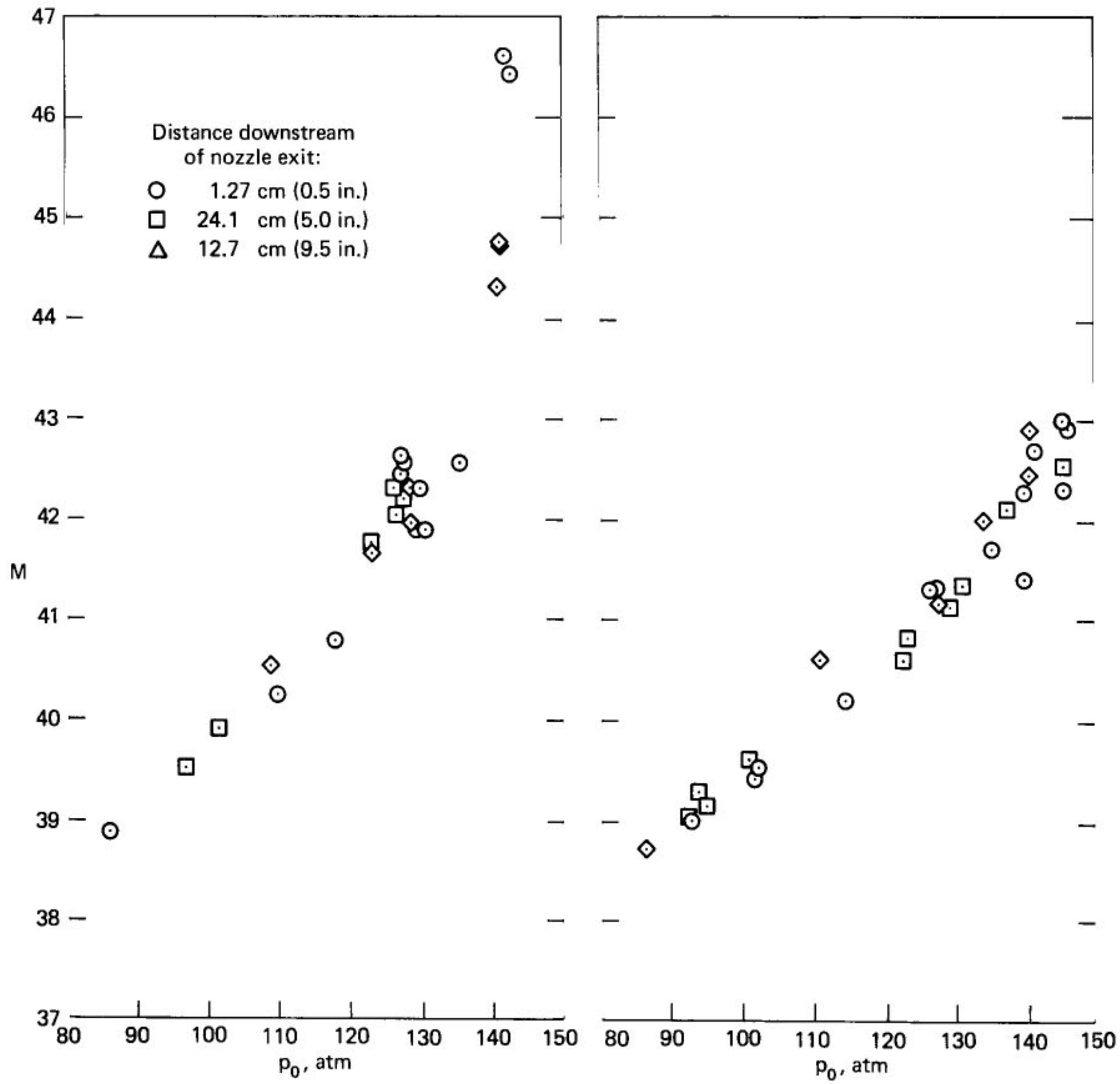


(b) p_o = 1880 psia; T_n = 1210° R



(c) p_o = 1640 psia; T_n = 1190° R

Figure 13.- Measured pressure distribution for the Mach-40 throat of a station 9.5 inches downstream of the nozzle exit.



(a) T_n = 1120° R

(b) T_n = 1180° R

Figure 14.- Variations in the test core Mach number with total pressure for the Mach-40 throat.

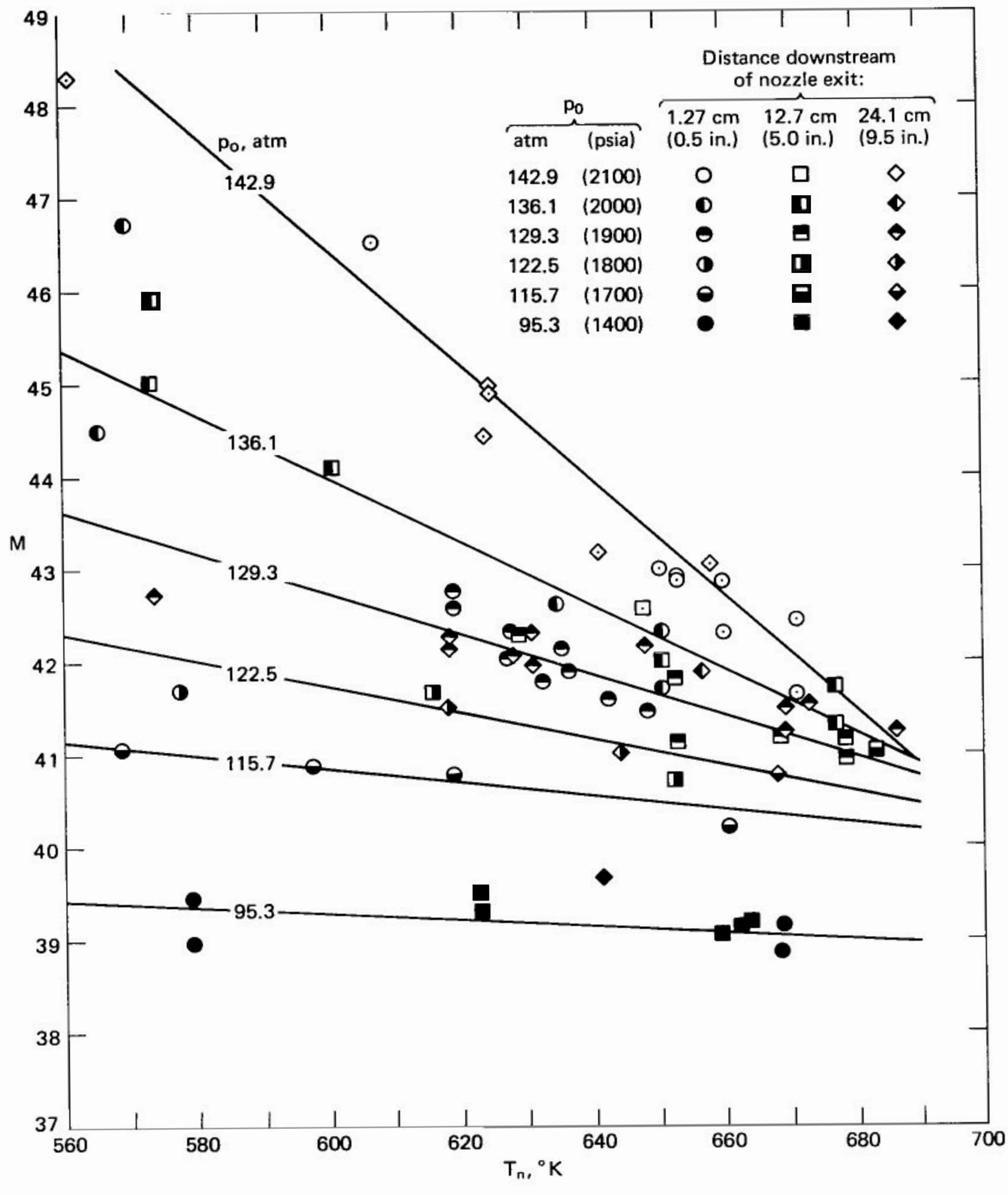


Figure 15.- Variations in the test core Mach number with nozzle temperature for the Mach-40 throat.

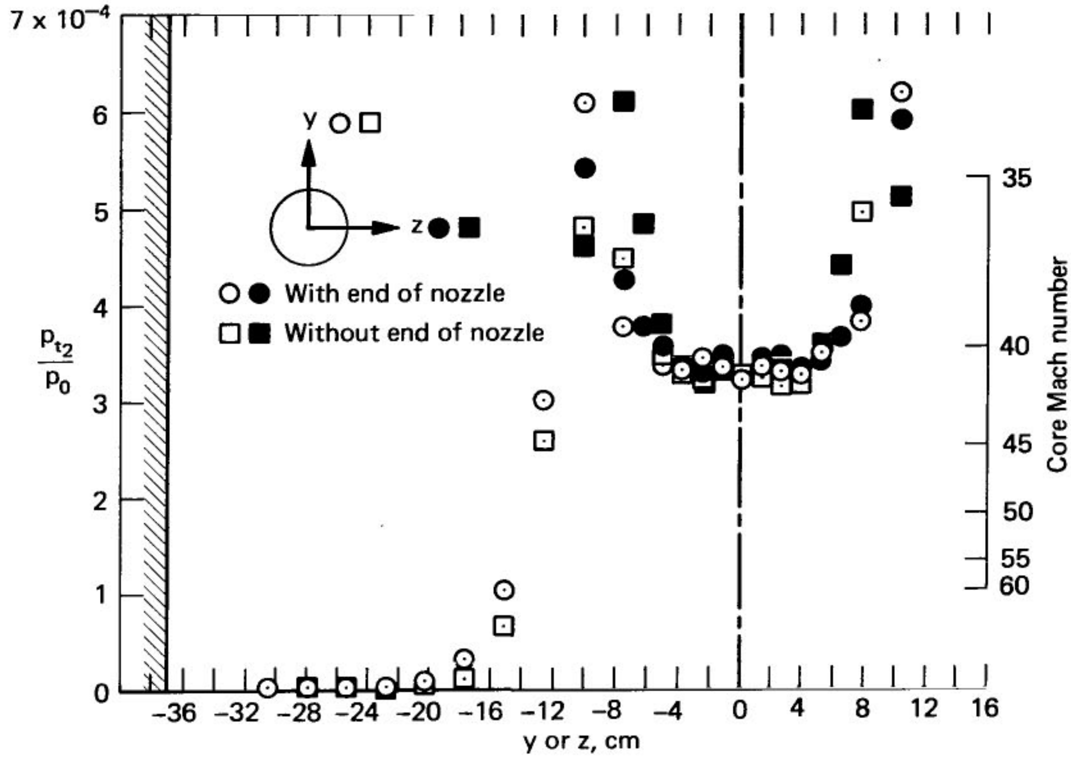


Figure 16.- Effect of shortening the nozzle on the flow characteristics for the Mach-40 throat; x = 0.5.

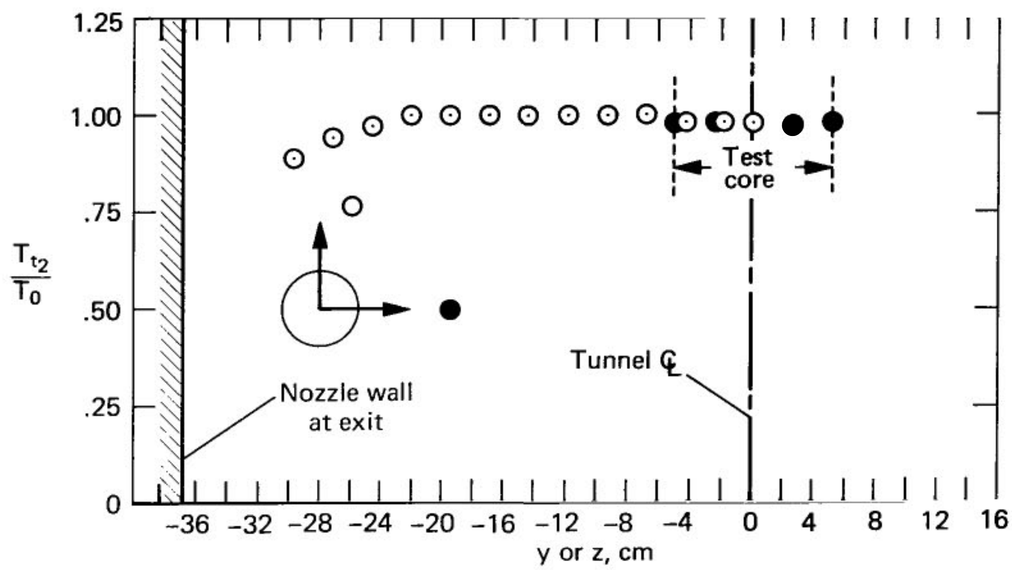
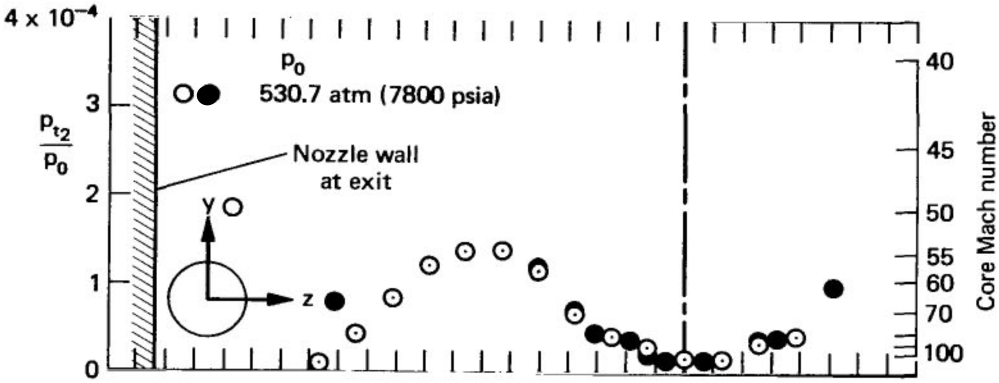
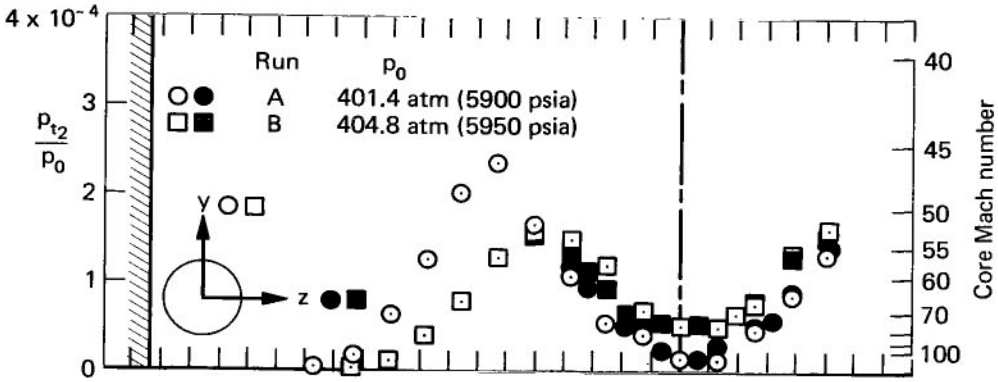


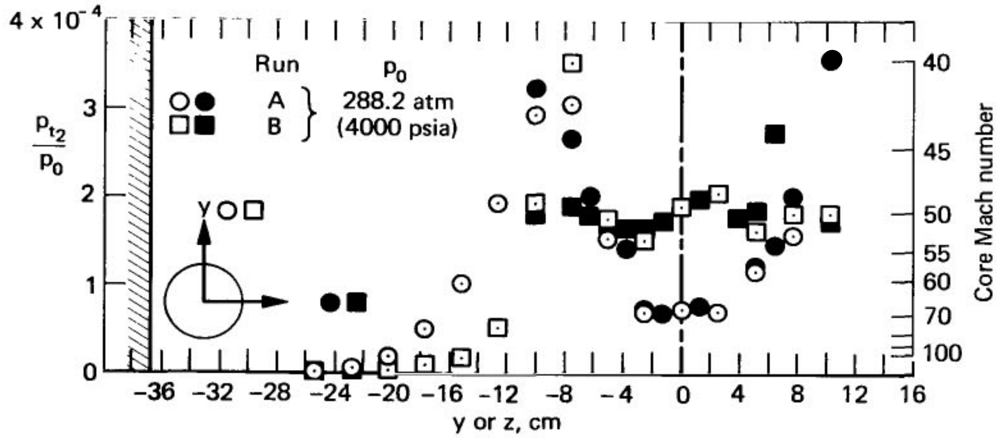
Figure 17.- Total temperature distribution for the Mach-40 throat at 0.5 inch from the nozzle exit; T_o = 1570° R.



(a) p_o = 7800 psia



(b) p_o = 5900 psia



(c) p_o = 4000 psia

Figure 18.- Pressure distribution for Mach-50 throat at 0.5 inch downstream of the nozzle exit.

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— NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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